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IN-SPACE TRANSPORTATION FOR GEO SPACE SOLAR POWER SATELLITES

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ABSTRACT

Space solar power satellites have the potential to provide abundant quantities of electricity for use on Earth. One concept, the Sun Tower, can be assembled in geostationary orbit from pieces transferred from Earth. The cost of transportation is one of the major hurdles to space solar power. This study found that autonomous solar-electric transfer is a good choice for the transportation from LEO to GEO.

INTRODUCTION

The goal of this study was to examine the transportation of space solar power (SSP) elements from low Earth orbit (LEO) to the operational orbit, geostationary Earth orbit (GEO). The effort of this study continued and built on work on SSP transportation at Boeing performed in 1998.[1] One of the findings in 1998 was that a rocket two-stage-to-orbit (TSTO) reusable launch vehicle (RLV) could deliver payloads to low Earth orbit (LEO) for a recurring cost of about \$370 / kg with a highly advanced rocket engine. This study used that information to help evaluate in-space transportation options. This study of in-space transportation for deployment of huge SSP satellites presumes that relatively small segments (<50,000 kg) are individually transported from low Earth orbit (LEO) to geostationary Earth orbit (GEO), where they are assembled into the large satellite, probably robotically.

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CONCEPTS

In this work,[2] a reference transportation system was first developed and analyzed. The reference concept used autonomous solar electric propulsion from LEO to GEO, as illustrated in Fig. 1. This point of departure design specifies autonomous transfer of each spacecraft segment that is transferred to GEO to make up the SSP Satellite. In this option, the launch vehicle places the payload and propulsion package in a 300 km equatorial LEO. The photovoltaic solar arrays are partly deployed to provide power for transfer. A cluster of Hall thrusters, using krypton propellant, move the payload to GEO. Vehicle thrust-to-weight (T/W) at LEO departure is very low, on the order of 10^{-4}. The cluster of Hall thrusters produce on the order of 100's of Newtons of thrust. The thrust arcs are continuous, except during passage through Earth's shadow. Once at GEO, the package rendezvous with, and attaches to, the existing partly-assembled Sun Tower satellite. There, the solar arrays are completely deployed, and the element takes its place as one of hundreds of similar array elements that produce power over the life of the SSP Satellite. The electric thrusters, no longer needed for transfer, can be used in a station-keeping mode until excess propellant is depleted. LEO to GEO spiral-out times were limited to 90 days as an estimate of a reasonable time to delay useful output from the investment.

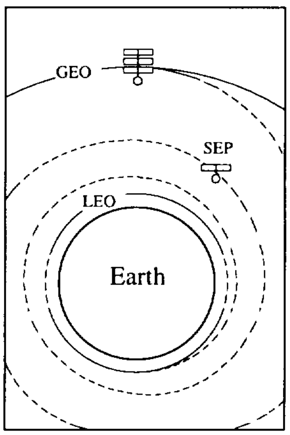

Figure 1. Illustration of autonomous solar-electric transfer from LEO to GEO.

In addition to the reference concept, several alternate in-space transportation options were evaluated. Figures 2-6 show the concepts considered. These are:

- First, a reusable orbital transfer vehicle (OTV) with chemical propulsion was considered with and without an aerobrake for aerodynamic capture back into LEO.

-Second, a reusable OTV with solar thermal propulsion.

- Third, the reference SEP autonomous transfer option coupled with a small initial boost from a tether. The tether also provided the deorbit impulse to the launch vehicle.

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- Fourth, a launch vehicle with less than orbital capability was considered with a pop-up stage. This small expendable chemical stage propels the payload to LEO. This strategy reduces the amount of Launch vehicle propellant, allowing for more payload mass, which in this case includes the mass of the pop-up stage.

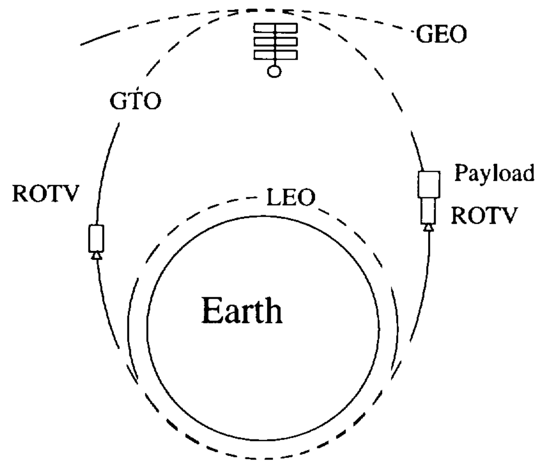

Figure 2. Reusable OTV concept.

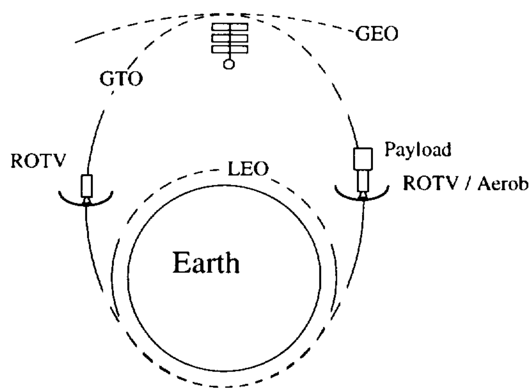

Figure 3. Reusable OTV with aerobrake concept.

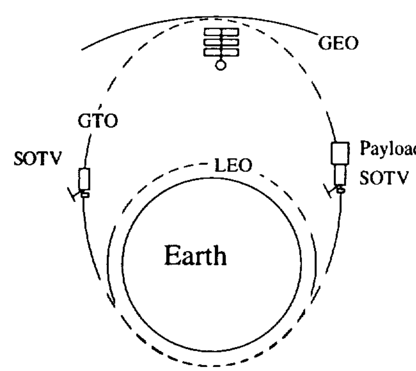

Figure 4. Reusable OTV with solar-thermal propulsion.

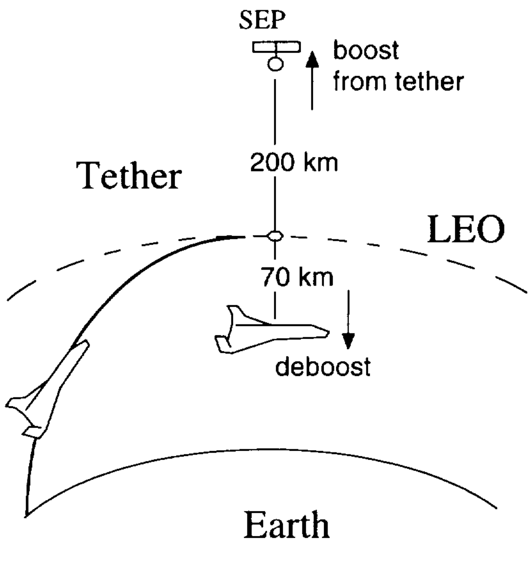

Figure 5. Concept using a tether boost.

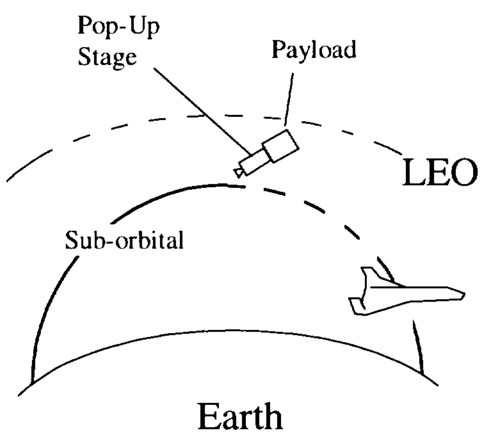

Figure 6. Concept with sub-orbital launch.

DESIGNS

The autonomous solar-electric concept is shown in Fig. 7 during the flight from LEO to GEO, with the solar arrays partially deployed to minimize radiation damage. The thruster arrangement is shown in Fig. 8. The concept used Hall thrusters. Ten thruster assemblies of 50 kW each are needed to provide transfer in 90 days, which was selected as reasonable for this study. Results of the design effort are shown in Fig. 9. All of the designs were configured to provide the same useful payload to GEO. An initial mass in low Earth orbit of 27,000 kg was selected as compatible with likely launch vehicles. The low Earth orbit selected was an altitude of 300 km, circular, and equatorial. Hall thrusters with direct drive were selected to avoid the need for power processing units. Some mass carried to GEO for the propulsion system was left attached during the useful life of the satellite but was not counted as useful payload. Included in that mass was tanks, thrusters, and the portion of the solar arrays destroyed during the transfer through the radiation belts. The degraded array mass, 1053 kg, represents the extra mass that must be provided so that the arrays provide the desired output at GEO. Figure 10 shows how the radiation belts degraded the solar arrays.

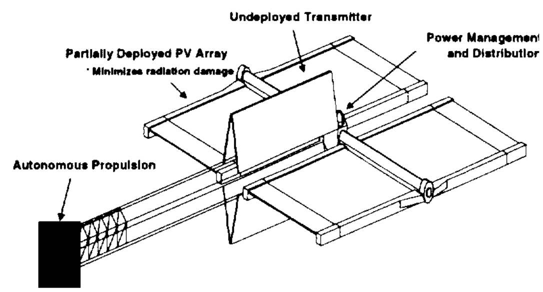

Figure 7. Sketch of autonomous transfer.

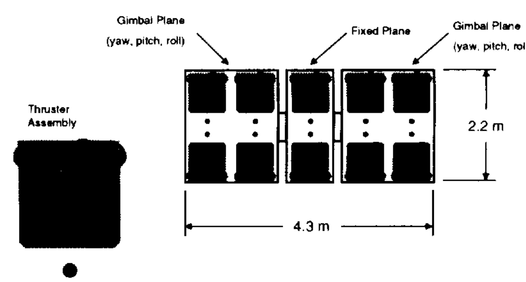

Figure 8. Single Hall thruster and bank of ten thrusters as used for autonomous transfer.

- Equatorial launch
- Partial array deployment for transfer
- No PPU, direct drive from high-voltage array
- Specific impulse 2 000 s
- Initial mass in LEO 27 000 kg
- Useable propellant 5 814 kg
- Mass in GEO 21 186 kg
- Residual propellant 174 kg
- Propulsion inert 2 810 kg
- Solar array degraded 1 053 kg
- Useful payload 17 149 kg

Figure 9. Design results for autonomous transfer.

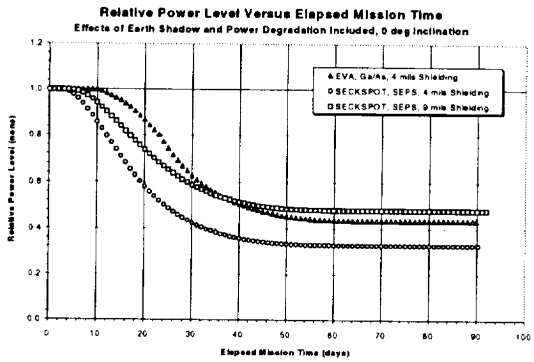

Figure 10. Degradation of solar arrays by radiation belts.

The reusable OTV is shown in Fig. 11, and the design results are shown in Fig. 12. The ROTV has a lifetime of 200 flights. An oxygen and hydrogen chemical rocket engine is used. The payload, ROTV, and propellant are at a node in LEO prior to departure. After delivering the payload, the ROTV departs from GEO into a geosynchronous transfer orbit (GTO) and returns to LEO using chemical propulsion. The relatively low specific impulse of chemical propulsion, even with hydrogen and oxygen propellants, leads to a high initial mass.

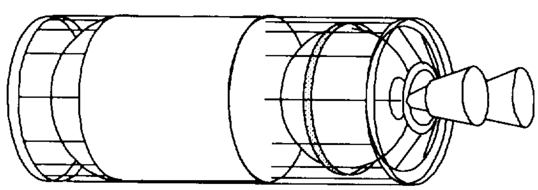

Figure 11. Reusable OTV.

- Equatorial launch
- Initial T/W 0.4
- Initial mass in LEO 86 053 kg
- Useful propellant 60 009 kg
- Reserves, resid., RCS 1 143 kg
- Stage inert 7 752 kg
- Payload 17 149 kg
- Specific impulse 470 s

Figure 12. Design results for reusable OTV.

The reusable OTV with aerobrake is shown in Fig. 13, and the design results are shown in Fig. 14. The engines fire through doors in the aerobrake. The aerobrake reduces the propellant needed for slowing into LEO; only a small circularization burn is needed to return to LEO. Because this propellant would be carried for the entire mission, it has a significant impact on the initial mass.

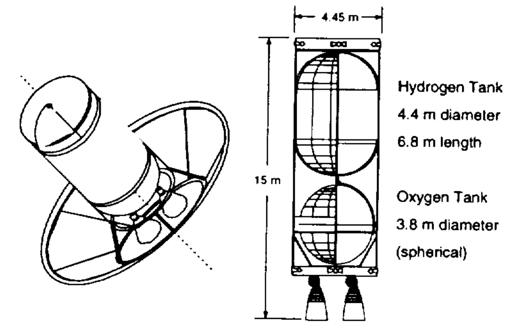

Figure 13. Reusable OTV with aerobrake.

- Equatorial launch
- Initial T/W 0.4
- Initial mass in LEO 62 274 kg
- Useful propellant 38 113 kg
- Reserves, resid., RCS 748 kg
- Stage inert (less aeroshell) 5 312 kg
- Aeroshell 952 kg
- Payload 17 149 kg
- Specific impulse 470 s

Figure 14. Design results for reusable OTV with aerobrake.

Figure 15 shows the effect of various assumptions for the aerobrake mass on the stage mass. The results shown above are based on the estimate that the aerobrake mass will be 15% of the mass braked.

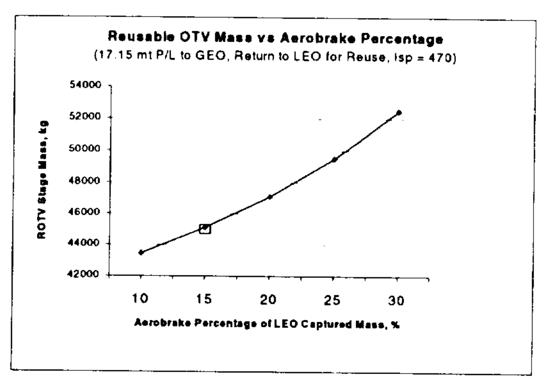

Figure 15. Effect of aerobrake mass on stage mass.

An artist's concept of a solar-thermal propulsion system and stage is shown in Fig. 16 as it might appear in an early technology demonstration mission. The reusable OTV with solar-thermal propulsion is shown in Fig. 17, with the design results in Fig. 18. Because the thrust of the SOTV is low, the trajectory actually has many thrust periods. The actual trajectory is more complicated than the one shown in Fig. 4 and employs multiple perigee and apogee thrust impulses. The thrust arcs are optimized to minimize finite thrust losses (delta-V) while providing a reasonable transfer time. The specific impulse of the solar-thermal propulsion is higher than for chemical propulsion, and the initial mass is reduced, but it is still higher than for the autonomous solar-electric approach.

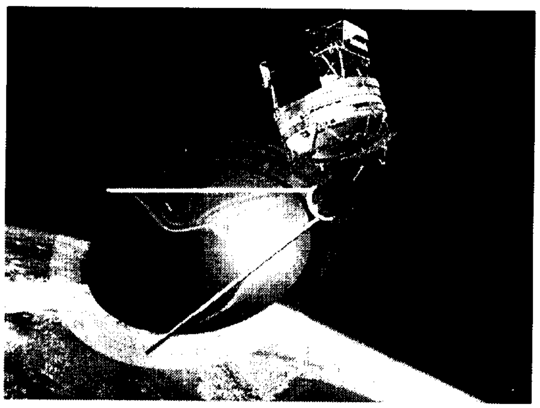

Figure 16. Artist's concept of solar-thermal propulsion system and stage.

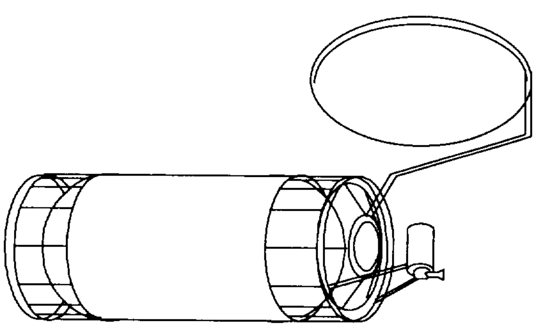

Figure 17. Sketch of reusable OTV with solar-thermal propulsion.

• Equatorial launch	
• Initial T/W	0.00016
• Initial mass in LEO	52 877 kg
• Useful propellant	25 634 kg
• Resv., resid., b/o, RCS	797 kg
• Stage inert	9 297 kg
• Payload	17 149 kg
• Specific impulse	881 s

Figure 18. Design results for the reusable OTV with solar-thermal propulsion.

Figures 19 and 20 give characteristics and design results of the tether option. The design was a relatively conservative tether, with no rotation and no electrodynamic acceleration. The change from the baseline autonomous approach was minimal, with small benefits to the solar-electric transfer and additional benefits to the launch vehicle. The tether boosts the payload more than 450 m/s, which reduces the mass in LEO about 3.5%.

While the reduction in mass and cost provided by the tether is not large, it is an indication of the potential of tethers. Further reductions from more aggressive tether options should be considered but were beyond the scope of the current effort.

- Launch vehicle docks at LEO node
- Launch vehicle is lowered on tether as payload is raised on tether
- Center of gravity stays at LEO orbit of 300 x 300 km
- Simultaneous release of launch vehicle and payload
- No disturbance of LEO node orbit
- Tether length down limited to avoid heating and drag
- Tether length up limited to 200 km

Figure 19. Characteristics of tether design.

• Equatorial launch	
• Partial array deployment for transfer	
• No PPU	
• Specific impulse	2 000 s
• Initial mass in LEO	26 044 kg
• Useable propellant	5 133 kg
• Mass in GEO	20 911 kg
• Residual propellant	154 kg
• Propulsion inert	2 605 kg
• Solar array degraded	1 003 kg
• Useful payload	17 149 kg

Figure 20. Design results for tether option.

Some characteristics of the sub-orbital launch option are shown in Fig. 21. The transfer to GEO is identical to the autonomous design, and the results in Fig. 9 apply. There was a benefit to the launch vehicle which will be discussed later.

The analysis of the sub-orbital option was based on the assumption of a two-stage fully-reusable rocket launch vehicle. Such a vehicle is quite likely to be a good selection for SSP launches. The sub-orbital option would increase the potential of single-stage launch vehicles, which could potentially reduce costs.

• Autonomous transfer LEO to GEO	
• Launch vehicle releases payload ~300 m/s short of LEO	
• Smaller launch vehicle, semi-global glide	
• Expendable "pop-up" stage propels payload to LEO	
• Oxygen/kerosene propulsion	
• Payload to LEO	27 000 kg
• Stage gross mass	3 000 kg
• Stage propellant	2 700 kg
• Stage inert	300 kg

Figure 21. Characteristics of the sub-orbital launch option.

Figure 22 introduces two additional concepts that were examined. An attempt to improve on the Hall thruster of the baseline option resulted in a higher specific impulse. A hybrid option was briefly considered which used a reusable OTV with aerobrake (ROTV-AB) for only a small portion of the velocity increment, and autonomous solar-electric propulsion completed the transfer. Figures 23-26 show the results. In designing the ROTV-AB for the hybrid option, the results improved significantly when the payloads were carried two at a time. For designs with a larger velocity increment from the ROTV-AB, grouping the payloads may not be needed.

- •Improved Hall Effect Thruster
  - •4000 second Isp
- •Hybrid System
  - •Reusable Aerobraked OTV
    - •Supplies part of the initial delta V for transfer
  - •Autonomous transfer from intermediate trajectory to GEO

Figure 22. Additional concepts considered.

• Equatorial launch		
• Partial array deployment for transfer		
• No PPU, direct drive from high-voltage array		
• Specific impulse	2 000 s	4 000 s
• Initial mass in LEO	27 000 kg	24 632 kg
• Useable propellant	5 814 kg	2 797 kg
• Mass in GEO	21 186 kg	21 835 kg
• Residual propellant	174 kg	84 kg
• Propulsion inert	2 810 kg	3 216 kg
• Solar array degraded	1 053 kg	1 386 kg
• Useful payload	17 149 kg	17 149 kg

Figure 23. Design results for improved thrusters compared to baseline results.

- Equatorial launch
- ROTV-AB provides transfer of 2 payloads to 300 km x 9000 km orbit
- Autonomous transfer of each payload to GEO
- ROTV aerobrakes directly from the 300 km x 9000 km orbit

Figure 24. Characteristics of hybrid concept.

• Partial array deployment for transfer	
• No PPU, direct drive from high-voltage array	
• Specific impulse	2 000 s
• Init. mass at 300 x 90000	23 138 kg
• Useable propellant	3 643 kg
• Mass in GEO	19 495 kg
• Residual propellant	109 kg
• Propulsion inert	1 684 kg
• Solar array degraded	552 kg
• Useful payload	17 149 kg

Figure 25. Design results for autonomous portion of hybrid option.

- Initial T/W 0.4
- Initial mass in LEO 68 658 kg
- Useful propellant 18 103 kg
- Reserves, resid., RCS 392 kg
- Stage inert (less aeroshell) 3 296 kg
- Aeroshell 591 kg
- Specific impulse 470 s
- Data for 2 payloads

Figure 26. Design results for ROTV-AB portion of hybrid option.

COMPARISONS

The initial mass in LEO for each of the concepts is compared in Fig. 27. The most obvious result is that the options with lower specific impulse have higher propellant requirements. Even the hybrid option has considerably more propellant than the reference autonomous SEP option. The option with the improved Hall thruster (4000 sec Isp) has reduced propellant mass. The inert mass is also larger for the reusable OTV options; solar-thermal has the highest inert mass.

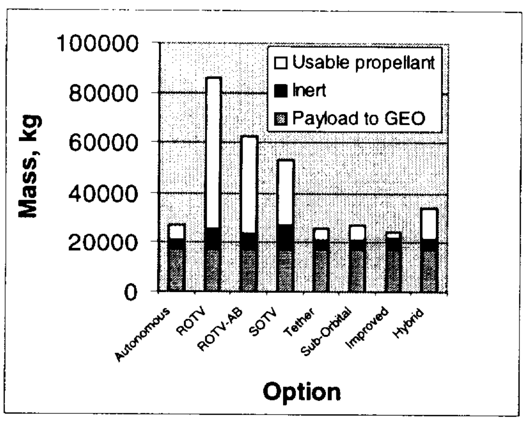

Figure 27. Comparison of mass in LEO of several in-space transportation options.

More important than the mass results are the cost results shown in Fig. 28. The cost of transporting propellant to LEO leads to high costs for the reusable OTV cases. The hybrid option, however, is not much different from the baseline in cost. Not much effort was given to the hybrid option, and it is not optimized, and it should therefore be considered competitive. The tether option reduced costs minimally, but no costs were included for operation of the tether.

The sub-orbital option showed some benefit to the launch vehicle. Unfortunately, the cost of the pop-up stage, which was expendable, offset the benefits. By integrating the pop-up stage into the payload, some savings might be possible compared to a separate vehicle. The operations of the launch vehicle are a problem for this concept, because the orbiter must return to a down-range site. There is a possibility that a semi-global site cold be used, with two flights returning the orbiter to the original site.

One interesting result is the "improved" Hall thruster. With the higher specific impulse, more power is required to provide enough thrust to complete the transfer in 90 days, which was the assumed requirement for this study for electric propulsion vehicles. Additional performance degradation occurred to the larger exposed arrays. As a result, the savings in launch costs because of the higher specific impulse were more than offset by higher transfer costs.

A goal of the study was to provide transportation to GEO at $800 / kg. The initial assumption was that the cost would be split between launch at $400/kg and transfer at $400/kg. As mentioned earlier, the cost analysis assumed $370/kg for the launch costs, based on Ref. 1. The total costs for the autonomous option are slightly over $800/kg. These costs are split into launch of the payload at $370/kg and the cost of transfer, which includes the cost of launch of the transfer propellant and transfer stage inert mass. The cost of transfer is somewhat over the goal of $400/kg. Note that the costs mentioned are only recurring costs and do not include development and business aspects.

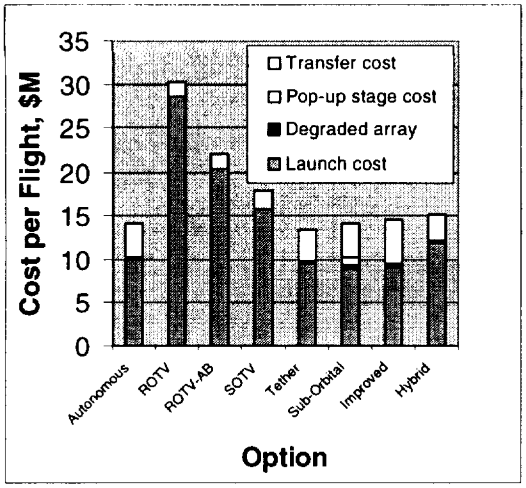

Figure 28. Comparison of recurring cost per flight of several in-space transportation options.

CONCLUSIONS

The results of this study lead to the following conclusions:

1. Autonomous solar-electric propulsion from low earth orbit to geostationary orbit is a reasonable choice for the Sun Tower solar power satellites.

3. The cost of transportation is likely to be close to the goal of $800/kg with the concepts studied, given the assumptions made in this evaluation.

ACKNOWLEDGEMENT

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1. The Boeing Company. *Space Solar Power End-to-End Architecture Study: Final Report*, NASA Marshall Space Flight Center contract number NAS8-98244 (Jan. 15, 1999).

2. The Boeing Company, *In-Space Transportation for GEO Sun Tower*, Mod 3 to NASA Marshall Space Flight Center contract number NAS8-98244 (22 December, 1999).